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QUARTERLY PROGRESS REPORT

Contract NAS8-35339

PLASMA SOURCE FOR SPACECRAFT POTENTIAL CONTROL

July 1983 - September 1983

by

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INTRODUCTION

The first quarter of the contract period was marked by substantial progress in all areas of the contract. It is anticipated that all of the assigned tasks will be completed within the contract period within the allocated budget. Progress in each area is as follows:

1. Informational Meeting with P. Wilbur

Dr. Paul Wilbur presented basis details of the hollow cathode operation to the plasma source team. The meeting was attended by C. R. Chappell (NASA/MSFC), Dr. Karl Knott (ESA/ESTEC), Dr. Arne Pedersen (ESA/ESTEC), and R. C. Olsen (UAH). The fuel requirements were established along with preliminary estimates of power and control requirements.

2. Spacecraft Interface Document (SID)

Using the material developed in the meeting with Paul Wilbur, a preliminary SID was developed and sent to the OPEN project office. This material was also presented at an OPEN Science Working Group (SWG) meeting (see below). A copy of this document is included as part of the quarterly report.

3. Preliminary Schematic

A preliminary schematic has been developed and included in the first draft of the SID (see above). The issue of coupling to the electric field experiment has not been resolved, but will be shortly.

4. Fuel Supply

The issue of how to store the xenon fuel supply - liquid versus gas - has been settled in favor of a gas supply for the moment. A complete fuel assembly - tank, valve, and regulator - has been designed by Moog Industries for a nearly identical task for the Air Force. Substantial cost

savings should be realized by adopting their design intact. This design was incorporated into the SID (see item 2).

5. Plasma Density Profiles

Paul Wilbur had conducted a short series of tests at Colorado State University on the nature of the xenon plasma distribution emitted from the hollow cathode. These data are currently being plotted and this information will be presented by the end of the contract.

6. Computer Modeling

A sequence of NASCAP models were run on a spacecraft modeled after the DE satellite with a 5 eV ion beam. This preliminary model showed all of the ions leaving the satellite, with none returning unless the satellite charges to the beam energy. Further modeling will be done to optimized the source location and determine the nature of returning ion fluxes.

MEETING AND PRESENTATIONS

The preliminary design for the plasma source (PSPoC) was presented to the OPEN Science Working Group (SWG) at their last meeting. The design was accepted, and a vote taken on adding the PSPoC to the OPEN payload. The plasma source was accepted for the EML and PPL satellites and may be added to the GTL and IPL satellites.

The plasma source design was presented at the 17th ESLAB symposium on Spacecraft-Plasma Interactions and their Influence on field and particle measurements, along with papers demonstrating the need for such a device. The American and European scientists at the meeting agreed there is a need for such a device on the OPEN mission.

REFERENCES

Olsen, R. C., D. L. Gallagher, C. R. Chappell, J. L. Green, and S. D.

Shawhan, A potential control method for thermal plasma measurements on the DE-1 spacecraft.

Olsen, R. C., P. M. E. Decreau, J. F. E. Johnson, G. L. Wrenn, A. Pedersen, and K. Knott, Comparison of thermal plasma observations on SCATHA and GEOS.

Pedersen, A., C. R. Chappell, K. Knott, and R. C. Olsen, Methods for keeping a conductive spacecraft near the plasma potential.

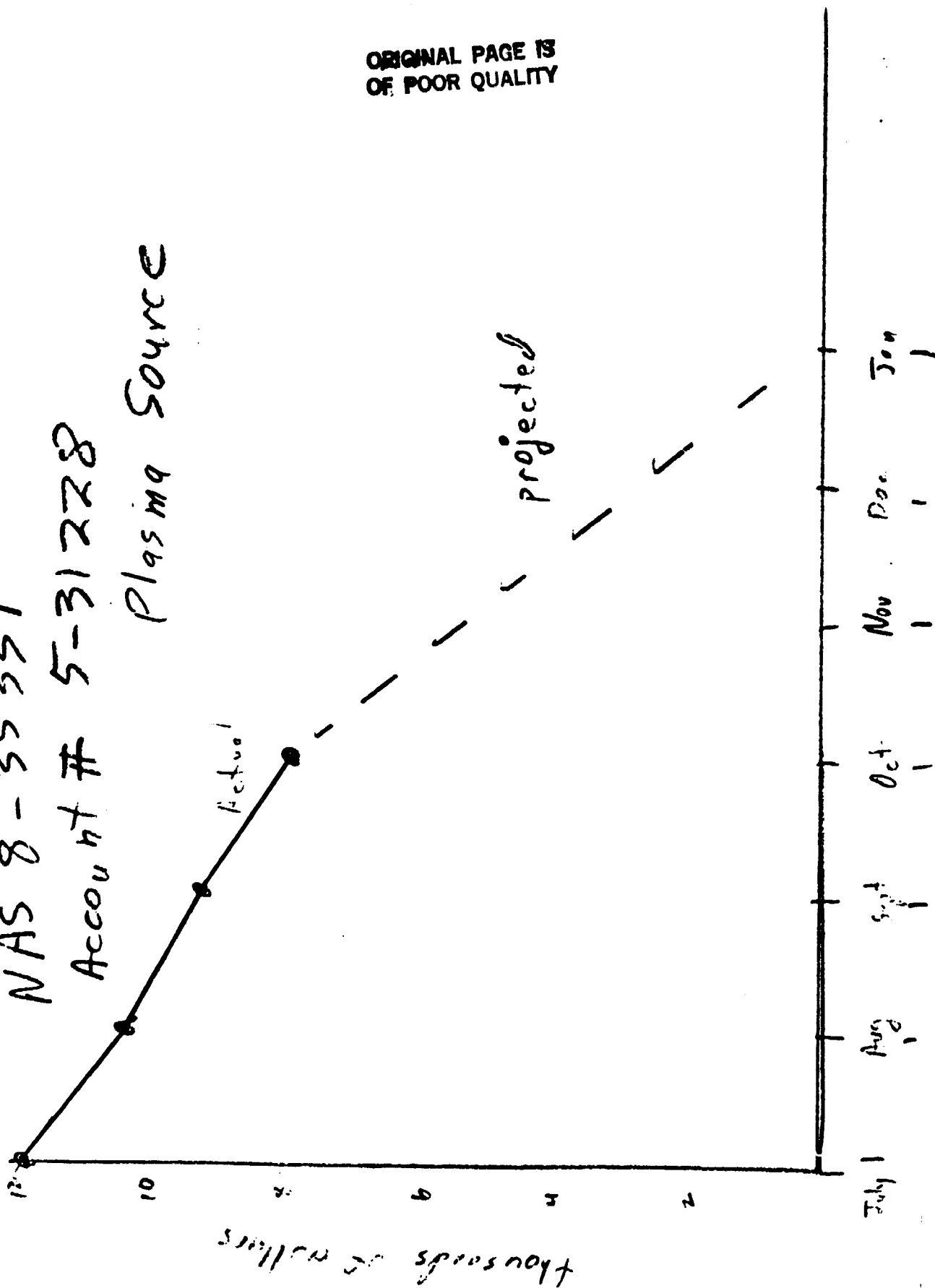
Whipple, E. C., R. C. Olsen, I. Krinsky, and R. Torbert, Anomalous high potentials observed on ISEE-1 in sunlight.

BUDGET

As of October 3, 1983, we had spent \$3,882.45 with \$7,885.79 remaining. At this rate, the contract will be completed within budget. A graph of contract spending is included.

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NAS 8-35339
Account # 5-31228
Plasma Source



OPEN REPORT

1.0 INTRODUCTION

Spacecraft in the earth's magnetosphere normally respond to the environment of ultraviolet light and ambient plasma by charging 1-20 V positive, when illuminated by the sun, and to potentials from a few volts to many thousands of volts negative when in shadow. Such potentials normally degrade the quality of the measurements by particle and field experiments. In particular, for a positive spacecraft, the ambient ion population with energy less than the spacecraft potential is hidden from view.

Large negative potentials on spacecraft have also been associated with spacecraft anomalies and failures. This is presumably due to differential charging over the spacecraft surfaces to potentials of kilovolts, with resulting arcs or discharges damaging surface materials and causing logic upsets in the spacecraft control circuitry.

For these reasons, it is desirable to control the spacecraft mainframe potential, and the potentials of any surfaces not directly connected to the mainframe (especially insulators). A plasma source is a nearly ideal way to provide such control. Experiments on Applied Technology Satellite 6 in 1974 and 1976 showed that plasma emission could discharge large negative potentials on the mainframe and all exposed surfaces. The hollow cathode plasma source, also known as a plasma bridge neutralizer when used with ion engines, worked well in sunlight and eclipse. It was also capable of reducing positive potentials from +10 V to ± 2 V. The resulting spacecraft potential was found to depend upon the potential of the plasma source (particularly the anode, or keeper). Control of the anode potential, therefore, provides the ability to control the spacecraft potential (Olsen, 1981).

Similar technology was implemented for an ion gun on the P78-2 satellite (SCATHA). This device utilized a hollow cathode, and xenon as the expellant. Experiments with this device in 1979 showed it behaved similarly to the ATS-6 neutralizer, when operated in a similar mode.

In order to control the spacecraft potential of the satellites in the OPEN mission, it is necessary to implement a device of this type. The

goals of the proposed Plasma Source for Potential Control (PSPoC) experiment are

- o To provide a stable electrical ground to enable the particle spectrometers to measure the low energy particle populations.
- o To measure the current required to neutralize the spacecraft.

In addition, the PSPoC will be able to prevent high charging events which could affect spacecraft electrical integrity. To fulfill these goals, the plasma source must be able to emit a plasma current large enough to balance the sum of all other currents to the spacecraft. In ion thrusters, hollow cathodes provide several amperes of electron current to the discharge chamber. PSPoC will be more than capable of balancing the net negative currents found in eclipse charging events, producing 10-100 microamps of electron current (Olsen, 1981). The largest current that will be required is the ion current necessary to balance the total photoelectric current. For a photocurrent density of 10 to 50 $\mu\text{A m}^{-2}$ (Grard, 1973), a spacecraft of 10 m^2 area requires a plasma source capable of 100 to 500 μA of ion current. The results of the ATS-6 ion engine operation (Olsen, 1981) and the laboratory results of Komatsu and Sellen (1978) show that a plasma source is also capable of the necessary ion current. The potential-clamping ability of the plasma source will provide an advantage that was not needed on the 3-axis stabilized ATS-6 satellite. A spinning satellite that is not azimuthally symmetric will fluctuate in potential as it rotates. The primary cause of such an effect is the variation in the photoemissivity of the spacecraft materials around the spacecraft. This effect was seen on the SCATHA satellite by the SC-10 Electric Field Experiment. The emission of a plasma will tend to damp out such potential excursions.

2.1 DESIGN GOALS

The goals of the PSPoC design, therefore are

- o To provide 0.5 milliamps ion current
- o To provide 0.1 milliamps electron current

- o To establish spacecraft potentials from +5 to -5 V, with ± 1 V accuracy
- o To measure the net current emitted by PSPoC from +500 microamps to -100 microamps with 1 microamp resolution
- o To provide a 5 year lifetime at 100 % duty cycle
- o To operate at less than 10 W steady state
- o To require no operator intervention after initial testing

2.2 PROPOSED DESIGN

In Figure 1, we show a block diagram of the principal elements of the PSPoC. These consist of the ion source, fuel supply, microprocessor controller, power supplies, and ammeters. A preliminary mechanical design is shown in Figure 2, illustrating the electronic package, fuel supply, and vacuum enclosure, which contains the hollow cathode. The principal features of the PSPoC are listed in Table 1. PSPoC is based on technology developed for the Satellite Positive Ion Beam System (SPIBS) flown on the P78-2 (SCATHA) spacecraft. PSPoC is designed to operate for the full five-year mission life at a duty cycle varying between 40 and 100%. In the discussion that follows, the engineering requirements considered in the design are reviewed and applied to the design of PSPoC.

TABLE 1

WEIGHT	5-7 kg
SIZE	10 x 10 x 50 cm
POWER	7W Steady State 10W Startup
TELEMETRY	~ 100 bits per second
EXPELLANT	Xenon
LIFETIME	20-40,000 hours
RESTARTS	2-10,000

2.2.1 ION SOURCE: HOLLOW-CATHODE DISCHARGE

The PSPoC concept is based on the establishment of a dilute plasma cloud that is composed of relatively low-energy electrons and ions. The ions in the plasma enable the transport of electrons through space, to neutralize charge imbalance, without requiring the establishment of high potential gradients. This is the concept on which the plasma-bridge neutralizer (used on ion thrusters) is based. These neutralizers have been demonstrated in space on the SERT II and ATS-6 ion thruster experiments. In these applications, a hollow-cathode discharge has been used to generate the plasma required for electron transport. Considerable development and qualification data have been accumulated for ion thruster hollow cathodes; consequently we propose to use a similar hollow cathode for PSPoC.

The hollow-cathode discharge, as operated for ion thruster neutralizers can provide electron and ion generation that is more than adequate to fulfill the PSPoC requirements. The elements required in a hollow-cathode discharge are shown schematically in Figure 3. Thermionic emission must be provided to ignite and maintain a "keep-alive" or keeper discharge between the internal surfaces of the hollow cathode and the keeper electrode. To minimize the heater power required, emission can be maintained at relatively low temperatures by employing barium to lower the cathode-surface work function. A bariated surface is provided by using a porous-tungsten-matrix reservoir or "insert" that is impregnated with barium aluminate. The insert forms the interior emitting surface in the hollow cathode discharge. By establishing appropriate operating conditions (heater power, discharge current, and expellant flow) a balance can be achieved between removal and replenishment of the surface barium thereby providing stable, long-lifetime cathode operation.

PSPoC development begins using hardware designs that are directly adapted from the SPIBS instrument. While the proven operation of existing hardware configurations is sufficient to meet all mission objectives, we believe that the following performance gains will be realized in PSPoC by incorporating relatively minor engineering changes:

- o Improvement of the cathode heater efficiency without degrading reliability, starting time, or cyclic lifetime capability.
- o Reduction of expellant flow requirement (hence: tankage mass) without increasing the keep-alive (keeper) discharge voltage.

2.2.2 FUEL SUPPLY

The fuel supply system is illustrated in Figure 4, with important parameters noted in Table 2. The straight-forward design shown here is adapted from the successful design used on SPIBS. The only change currently anticipated is an increase in the size of the fuel tank, and possibly a further reduction in the pressure at the regulator.

2.2.2.1 EXPELLANT

Xenon has been selected as the optimum choice for minimizing interactions with spacecraft systems and experiments while offering relatively high ionization efficiency. The mass of this ion is high enough to make it distinct from all of the ordinary magnetospheric ions (H^+ , He^+ , O^+), thus eliminating any possibility of interference with the ion mass spectrometers.

2.2.2.2 TANK

Approximately 2 kg of fuel are needed. The current design is for a 2-liter volume tank at 1000 psi, fed into an on-off valve and regulator (see below). The design uses off-the-shelf DOT-qualified tanks; the weight of the single tank (filled) should be approximately 3-4 kg including the manifold and extra tie-down straps required.

2.2.2.3 VALVE

The latching valve is solenoid operated, with positive latching. A 100 msec current pulse of about 1 A at 28 V is required to open the valve, and about .1 A is required to close the valve. The valve is rated at 1500 psi.

2.2.2.4 REGULATOR

The fuel tank will initially be filled to approximately 1000 psi. The source requires a gas pressure of a 5-10 psi at the porous plug. This need is met with an aneroid-type regulator using a sealed bellows movement. Outlet pressure can be adjusted over the 5-10 psia range to match the impedance of the plug.

2.2.2.5 POROUS PLUG

The final reduction in gas pressure occurs at the cathode where the pressure is reduced from a regulated 7 psia to a few torr by a porous plug. The plug is fabricated from tungsten with a density of 80%, E-beam welded into a tantalum housing.

TABLE 2

PARAMETER	VALUE
System Type	Regulated, high pressure
Reservoir	
a) Maximum operating pressure	1000 psi
b) Volume	2.0 liter
c) Typical capacity (Xe)	200 standard liters
d) Mass (filled)	2-4 kg
e) Material	TBD
Latching value	
a) Type	Solenoid-latching
b) Operating current (100 ns)	1A-open; 0.1 A-close
c) Mass	180 g
Pressure regulator	
a) Type	Aneroid
b) Outlet pressure	7 ± 0.3 psia
c) Outlet pressure adjustable range	5-10 psia
d) Minimum inlet pressure	20 psia
e) Mass	460 g
Pressure transducer	
a) Type	Semiconductor
b) Mounting	Built into a screw and attached to fill fitting

2.2.3 ELECTRONICS

The electronic design of the system is illustrated in Figure 5.

2.2.3.1 POWER PROCESSING

The power processing required for PSPoC provides power for the cathode keeper, cathode heater, neutralizer bias supplies along with the line regulator, and a.c distribution inverter. The PSPoC power utilization profile requires 10W at the time of cathode ignition, and 7W for nominal steady-state operation.

Assuming 24-28 V input, 400 mA are required for the cathode heater (at ignition), 80 mA for the discharge (steady state), and 40 mA for control and processing electronics.

Peak power requirements are at startup (ignition), lasting approximately 5 minutes.

Power converters are assumed to be from 20-40 kHz, adjustable to the master driving frequency.

Only one voltage supply exceeds 100 V; the discharge supply at ignition reaches approximately 200 V.

2.2.3.2 COMMAND AND CONTROL

The OPEN project office has stated their intention to provide a flight-qualified microprocessor for control purposes of the scientific packages. This processor will be implemented for the straight-forward ignition and steady state command sequences. The microprocessor will sort the PSPoC data stream, ordinarily passing small amounts of housekeeping telemetry, and the net current measurement which is the devices main output. Under special circumstances, the microprocessor will pass the housekeeping data at higher rates. We also anticipate a connection to the DC electric field measurement to provide a spacecraft potential monitor. This will be used to determine the PSPoC bias voltage, in order to set the potential of the spacecraft mainframe to near 0V.

Ordinary operation of the device will not require operator intervention, but initial testing will require commanding in an interactive mode. During initial startup and testing we will require approximately 10 commands per hour, with later operations requiring either no commands or approximately 1 command per orbit.

The instrument will have one command word of 8 bits, and 5 individual commands.

2.2.3.3 ELECTROMETER

The PSPoC design requires accurate measurement of the net current to the device over the 1-500 microamp range with $\pm 1\%$ accuracy. Furthermore, measurement of the net current is required to be bi-directional since the spacecraft net current may have either sign. Finally, the unit is required to have a low source impedance so that the voltage across the input terminals does not exceed ± 0.1 V at the highest input current (0.5 mA).

The current design utilizes amplitude compression at high current levels, and a straight-forward bipolar electrometer on the spacecraft side of the isolation circuitry.

2.2.3.4 TELEMETRY

The net current leaving PSPoC will be monitored with a bipolar electrometer having a range of ± 0.5 mA, requiring an analog channel with 4 8-bit words per second. PSPoC engineering data, i.e. keeper voltage, bias voltage, discharge current, and fuel pressure will be sampled at a very low rate, probably once per major frame (8 to 16 seconds). This will require 4 or 5 words of 8 bits. Digital channel requirements for monitoring instrument status are for 1 word of 10 bits sampled 4 times per second. Tables 3 and 4 illustrate the command and telemetry requirements currently anticipated.

2.2.4 LIFETIME

The lifetime requirements for the PSPoC instrument are not significantly different from those of a satellite-control ion thruster (15,000 to 30,000 hour operating time with 2000 restarts). Therefore, the use of ion thruster (neutralizer) technology should enable the achievement of PSPoC lifetime goals without requiring an extensive lifetime capability demonstration. Substitution of xenon for the usual ion thruster propellants already has been proven feasible in the SPIBS instrument and the effort required under the proposed program would be to verify that the hardware designs and operating conditions are sufficiently similar to those in SPIBS to make use of the available qualification and life-test results.

TABLE 3

TELECOMMANDSINDIVIDUAL COMMANDS

- Experiment on/off
- Gas supply open/closed
- Heater enabled
- E-field feedback on/off
- Execute Memory Load Telecommand

MEMORY LOAD COMMAND

- 1 word of 8 bits
 - o 5 bits to set 32 levels of Bias Voltage
 - o 1 bit to set Heater to manual or auto
 - o 1 bit to set discharge voltage high or low
 - o 1 bit to call for Rapid Duty Cycle

TABLE 4

TELEMETRYANALOG WORDS

1 word of 8 bit ~ 4 per sec; net current

4 words of 8-bit and low (~ 1 sec) sample time are required to transmit:

- Temperature of HC Heater
- Pressure in Gas Tank
- Bias Current of HC
- Discharge Voltage of HC

DIGITAL WORD

1 digital word of 10 bits is required. The word must be sampled a few times per second. This word will be super-commutated and the first two bits will give the commutation status.

Word 00: 8 bits for Bias Voltage

Word 01: 8 bits for Discharge Current

Word 10: Retransmission of Memory Load TC Word before execution

Word 11: Status Word composed of:

- 1 bit for Heater enabled
- 1 bit for Heater auto or manual
- 1 bit for Discharge 20 or 200V
- 1 bit for Rapid Duty Cycle on/off
- 1 bit for Gas Supply on/off
- 1 bit for Feedback on/off
- 1 bit for TEC execution signal
- 1 bit as spare

Hardware which has previously been flight-qualified will be used in all areas, where practical, in order to minimize qualification tests.

The five-year lifetime specified for PSPoC requires careful consideration in 4 areas. First, it will be necessary to scale the expellant system (tanks, valves, pressure reducers, etc.) to provide an adequate expellant supply for the time period required. Second the hollow cathode operating conditions will have to be adjusted to obtain an operating temperature of about 850° C to 950° C for the barium-impregnated, porous-tungsten cathode insert. This will ensure that depletion of barium from the insert surface is replenished from the interior of the insert. Third, the cathode heater thermal design will be revised to provide cathode ignition with less power input than has been typical for ion thruster or SPIBS cathodes. While this thermal redesign will consist of relatively minor changes, some verification testing will be required. Finally, long life reliability of the electronics package will have to be assessed and then validated by parts qualification.

2.2.5 INTECRATION: SPACECRAFT LOCATION

The ion source will be enclosed in a vacuum housing. This will enable us to test the device on the ground without requiring the entire spacecraft to be in vacuum. This will also protect the cathode and xenon supply from contamination. In space, the protective cover will be removed. Figure 6 shows the protective cover for the PSPoC instrument. This design has been used previously for the electron gun experiment on ISEE-1, and is flight and shuttle qualified. (Arends and Gonfalone, 1976)

The PSPoC will function most effectively if it is mounted on one of the ends of the spacecraft, with the discharge oriented parallel to the spacecraft spin axis. It should not be pointed towards the scan platform, and would preferably be removed from the other experiments by 10-50 cm. Figure 7 shows the preferred location on a spacecraft of the form anticipated for PPL, with scan platform.

3.0 OPERATION

The Plasma Source for Potential Control experiment will

- o Enable the particle experimenters to measure thermal ion and electron fluxes by clamping the potential near 0 V.

- o Measure the emitted current required to maintain the spacecraft near zero potential.
- o Prevent large potential build-up that might otherwise endanger spacecraft electrical integrity.
- o Reduce differential surface potentials on the spacecraft body.

3.1 POTENTIAL CONTROL FOR PLASMA MEASUREMENTS

The major use and importance of the PSPoC will be its use in enabling the particle spectrometers to measure low-temperature ion and electron fluxes. The measurement of thermal distributions is difficult without controlling potential. PSPoC therefore includes a small biasing capability to increase its utility in particle measurements.

The plasma current part of this experiment is the measurement of the current required to bring the spacecraft to a potential near zero. As has been demonstrated on SCATHA and the ATS-6 spacecraft, the emission of a neutral plasma is very effective in clamping the spacecraft to near zero potential. Plasma emission provides a current source of either sign depending on the potential difference between the spacecraft and the ambient plasma. As illustrated in Figure 8, a measurement of the current flowing from the PSPoC device is the total current required to neutralize the spacecraft. If the spacecraft only collected plasma currents, this measurement would be the net plasma current directly. However, since the spacecraft emits particles by secondary emission and other processes, the plasma current can only be inferred if the emission processes of the spacecraft and the incident flux energy distribution are known. Assuming a knowledge of the exterior of the spacecraft and data from the particle spectrometer measurements, the plasma current can be inferred.

3.2 INTERFERENCE

The emission of a particle flux from PSPoC will generate a return flux of particles visible to the particle detectors. This is not expected to be a significant problem because:

- o Particle detectors that can sense either mass or the incident angle of the incoming particles will be able to easily discriminate against the return flux of xenon.

- o The amount of xenon returning to the spacecraft away from the source is expected to be small.

The effects of PSPoC are not expected to hamper other instruments and in fact will benefit them by controlling differential potentials around the spacecraft and the absolute potential.

Relatively few data are available about high-frequency noise generated by the plasma discharge. Operations of the ion engine aboard ATS-6 did not generate signals that were detectable to the UCLA magnetometer. The time resolution for the instrument was $\frac{1}{2}$ sec., and sensitivity was about 0.1 gamma. No disturbances were being generated within these limits. Available laboratory measurements indicate that a full ion engine is somewhat noisy. Measurements of the electric field on SCATHA in the 0-6 kHz frequency range during operations of the SPIBS showed that there was little or no noise generated in that frequency range, for most environments. Magnetic loop data in the same frequency range showed substantial noise over most of that range. However, the PSPoC design eliminates both the discharge chamber and the high-voltage extraction electrodes. The discharge chamber with its large magnetic fields and the high voltage used in beam extraction are the main noise sources of an ion thruster. Measurements to characterize the noise spectrum and amplitude will be made in the development phase.

4.0

PSPoC MANPOWER

Table 5 shows the activities planned during the OPEN mission by the Plasma Source for Potential Control experiment team. During the definition phase, Dr. Olsen will coordinate with the other OPEN experimenters and detail the PSPoC program. Dr. Paul Wilbur will develop the source design, optimizing the hollow cathode design parameters. During the development phase, the hollow cathode and fuel supply will be built under the direction of Dr. Olsen, while the electronics systems will be built in Europe under the direction of Dr. Karl Knott. The subsections will be integrated in Europe under his direction. Dr. C. R. Chappell, principal investigator for the TIDE instrument, has the largest interest in the PSPoC effort of the Science Working Group, and represents the SWG and TIDE interests in the PSPoC group. The final phase will be the actual operation of PSPoC on the various spacecraft. Following an initial checkout period, the source will be operated according to the schedule determined by the OPEN experimenters.

5.0

DATA ANALYSIS

We will make use of the NASA-supplied OPEN data-handling system to carry out the data-reduction phase of our proposed program. Use of this facility will greatly facilitate required correlation of PSPoC data with data obtained by the particle and wave investigations. During PSPoC operation, Dr. Olsen will cooperate with the particle investigators in data analysis, and aid them in interpretation of the low energy plasma data taken during PSPoC operations.

T A B L E 5

PSPoC TEAM

R. C. Olsen (UAH)	Principal Investigator Coordination of activities, device modeling, fuel supply
K. Knott European Space Agency	Electronics design, assembly; instrument integration
P. Wilbur Colorado State University	Hollow cathode design
C. R. Chappell NASA/MSFC	SWG/TIDE coordination

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- 4) Olsen, R. C., "Modification of spacecraft potential by plasma emission," Journal of Spacecraft and Rockets, 18, 462, 1981.

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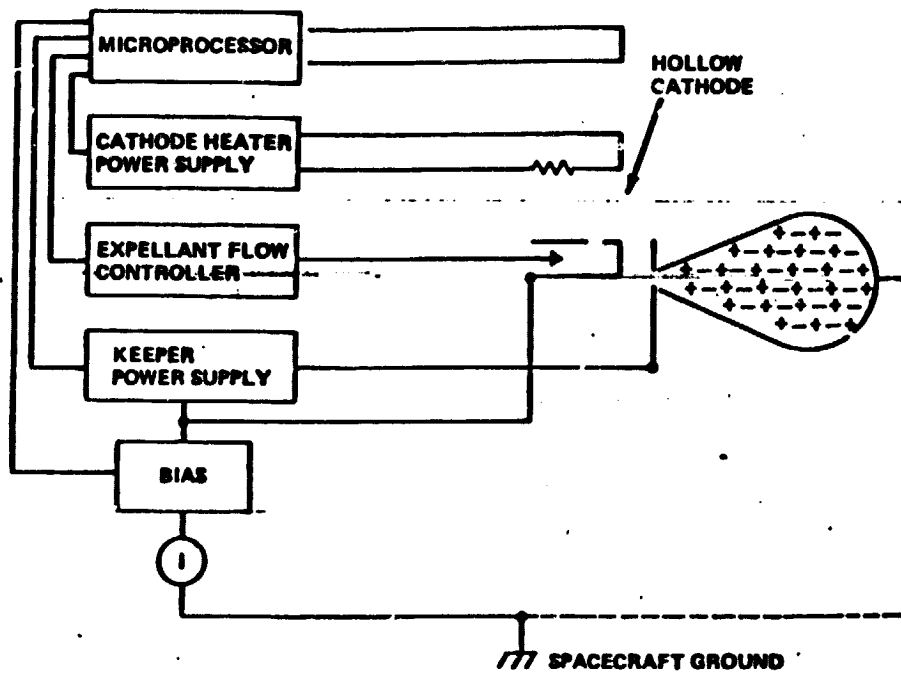


FIGURE 1. PSPoC BLOCK DIAGRAM

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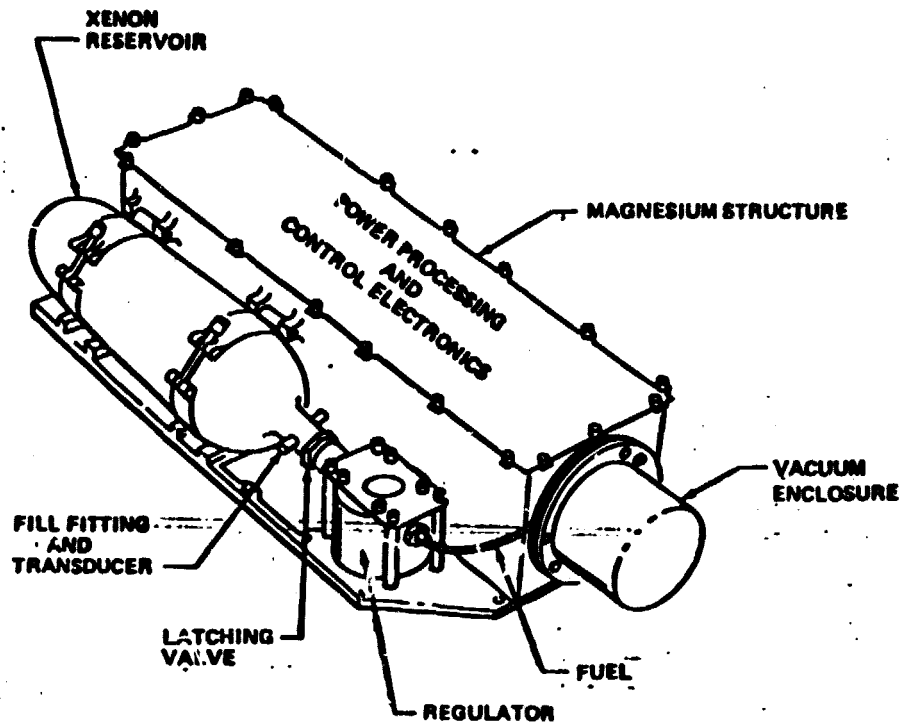


FIGURE 2. PSPoC MECHANICAL DESIGN

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HOLLOW CATHODE DESIGN

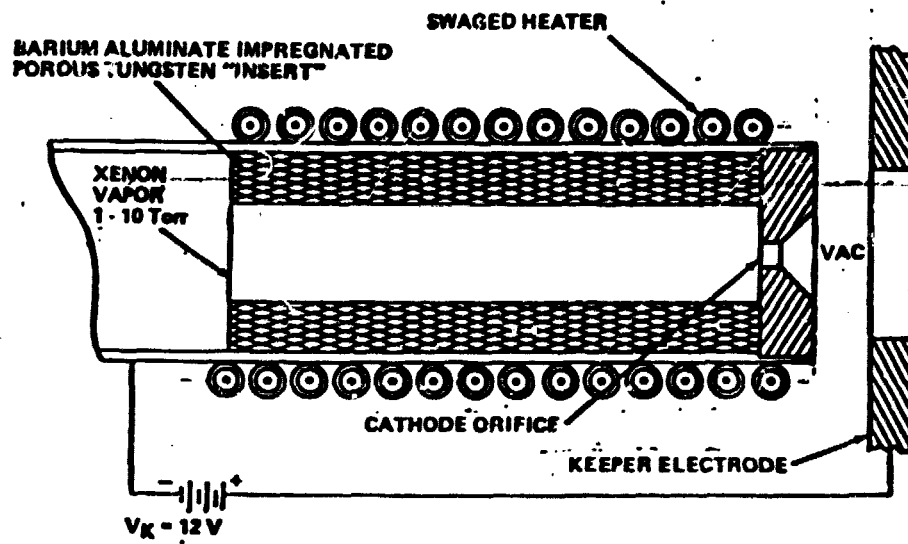


FIGURE 3.

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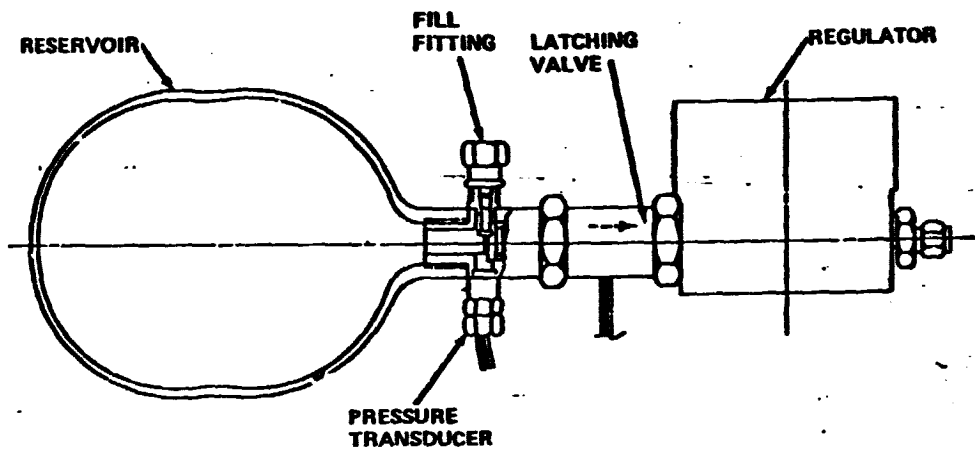


FIGURE 4. EXPELLANT ASSEMBLY LAYOUT DRAWING

ZERO ORDER SCHEMATIC OF PLASMA EMITTER



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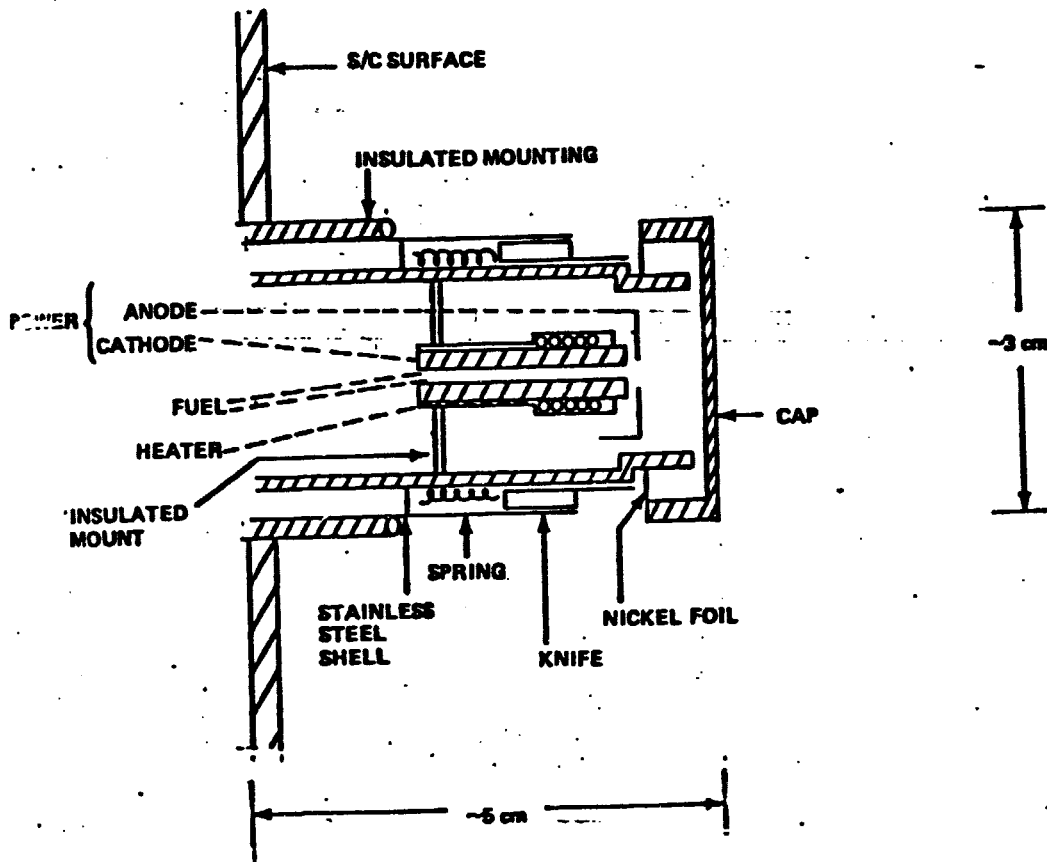


FIGURE 6. VACUUM COVER

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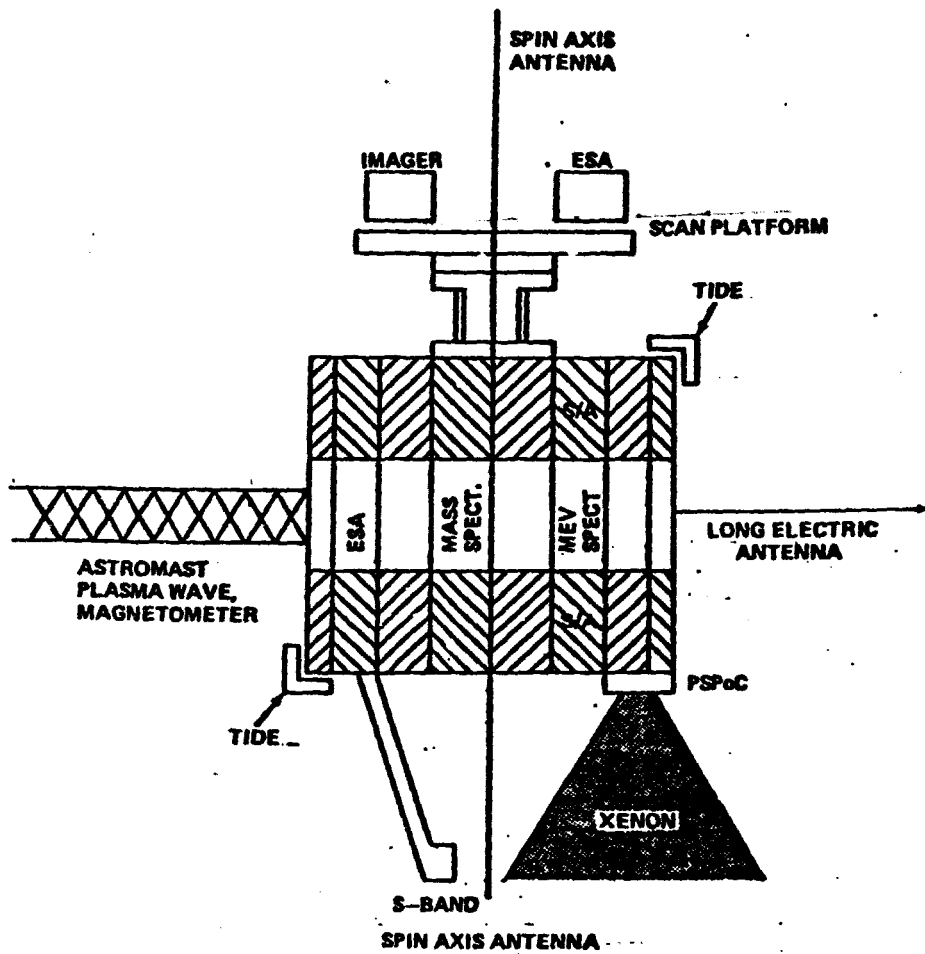
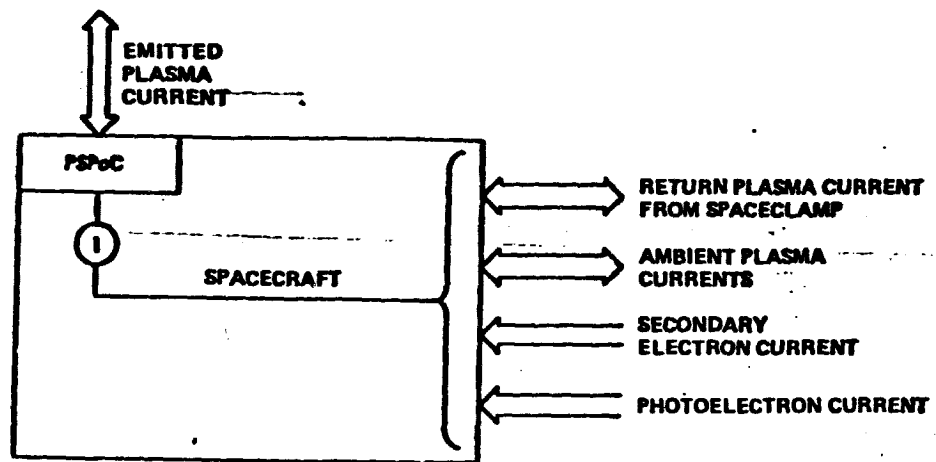


FIGURE 7.

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CURRENT BALANCE MEASUREMENT

FIGURE 8. CURRENT BALANCE OF OPEN SPACECRAFT